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NASA TM X-530

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TECHNICAL MEMORANDUM

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A PRELIMINARY STUDY OF SOME ABORT TRAJECTORIES INITIATED DURING LAUNCH OF A LUNAR MISSION VEHICLE

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

WASHINGTON

April 1961
Declassified February 6, 1962

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A PRELIMINARY STUDY OF SOME ABORT TRAJECTORIES INITIATED DURING LAUNCH OF A LUNAR MISSION VEHICLE*

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SUMMARY

An analysis was made of the abort capabilities of a circumlunar reconnaissance vehicle during three typical launch trajectories. The apogee, flight time, and entry conditions of abort trajectories initiated between near-orbital and escape velocities were evaluated with the use of elementary orbital mechanics and several abort velocities. In some of the cases the entire velocity was applied at the time of abort either to change the flight-path angle or to reduce the velocity. In other cases small velocities were applied at apogee to insure an acceptable atmospheric entry.

The trajectories obtained with abort velocities applied to change the flight-path angle were used to define recovery boundaries for various magnitudes of abort velocity. A recovery boundary was defined as the most extreme position along the launch trajectory from which the vehicle with a given abort velocity could return immediately to and/or remain within the earth's atmosphere. For abort trajectories initiated beyond the recovery boundaries, the results indicate that apogee distance and flight time were significantly shortened by reducing the vehicle velocity at abort. However, the procedure of delaying the velocity correction until the vehicle coasted to apogee required the smallest velocities to insure a return of the vehicle to the atmosphere.

In addition, several abort trajectories initiated at suborbital velocities and at relatively high flight-path angles were investigated. The trajectories, calculated from abort until the vehicle reached the surface of the earth, indicate that these conditions of relatively low-velocity and high flight-path angle could be critical from deceleration considerations.

INTRODUCTION

One of the most important requirements of the Apollo circumlunar mission will be the ability to abort the flight at any time and return to the earth safely. This requirement will be present not only during the actual mission but throughout the entire series of buildup or preliminary flights. Abort is, therefore, a phase of this mission which deserves early and considerable attention from both the National Aeronautics and Space Administration and industry.

A preliminary inspection indicates that the initial phase of a lunar trajectory may be divided into five regions during which the abort conditions have different aspects: (1) a region of low speed and low altitude which includes off-the-pad aborts, (2) a region of high dynamic pressure, (3) a region of relatively high flight-path angles at suborbital velocities while leaving the atmosphere, (4) a region between orbital and near-escape velocity, and (5) a postinjection coast period. Each of these regions may have different problems associated with a safe abort and may require different procedures, different thrust levels, and different thrust durations calling for more than one abort propulsion system.

In this paper a preliminary study is made of several possible methods of utilizing available fuel in the case of an abort during launch of a lunar mission. The study does not cover all of the previously mentioned regions but is limited to an investigation of the region between orbital and near-escape velocity (region 4) with a cursory study of the most critical abort conditions obtained at suborbital velocities beyond maximum dynamic pressure (region 3).

In these two regions an abort may be initiated deliberately (due to malfunction of the guidance system, etc.) or it may occur inadvertently (loss of thrust, failure of stages to separate, etc.). If boost-thrust cutoff can not be insured, the size of the abort rockets may have to be quite large to insure an initial separation of the vehicle from the booster. In any case, initial separation must be made away from the trajectory of the booster. Following this separation, the vehicle must be reoriented and in most cases a second thrust (or a continuation of the first) applied to change the velocity or flight-path angle or both. This second phase of the abort maneuver may be performed immediately or after some coast period. In this preliminary study only the second phase which involves the magnitude and direction of the applied abort velocity, the ensuing trajectories, and the conditions of entry into the earth's atmosphere will be studied.

SYMBOLS

In this paper, distances are measured in U.S. statute miles (1 U.S. statute mile = 1.60935 kilometers).

a_n — deceleration in direction of resultant aerodynamic force, g units

C_D — drag force coefficient, D/qS

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D — drag, lb

g — acceleration due to gravity, ft/sec^2

h — radial height above surface of earth, U.S. statute miles

I_{sp} — specific impulse, sec

L — lift, lb

q — dynamic pressure, lb/sq ft

r — radial distance measured from geographical center of earth, U.S. statute miles

S — vehicle effective drag area, sq ft

T — thrust, lb

t — time, sec

Δt — time from initiation of abort to entry, hr

V — velocity, ft/sec

ΔV — incremental vehicle velocity due to retro-rocket thrust, ft/sec

W — total weight of vehicle, lb

ΔW — weight of retro-rocket propellant, lb

α — angle of ΔV_o with respect to velocity vector (see fig. 4), deg

γ — flight-path angle, deg

Subscripts:

o values at initiation of abort

e values at entry into atmosphere

a values at apogee

p values at perigee

1 values of V_o and γ_o adjusted for effects of ΔV_o

max maximum value

abs absolute, denotes measurement with respect to inertial coordinates at the center of the earth

rel relative, denotes measurement with respect to rotating (24-hr period) coordinates at the center of the earth

ANALYSIS

Launch Trajectories

In the analysis, aborts initiated from the three typical launch trajectories shown in figure 1 are investigated. The two launch trajectories shown in figures 1(a) and 1(b) are for four-stage versions of the Saturn booster and the trajectory of figure 1(c) is for a three-stage version of the Saturn booster (C-2 version). In each figure the three quantities pertinent to this study are shown plotted against time. The altitude, relative velocity, and relative flight-path angle for the first two stages and altitude, absolute velocity, and absolute flight-path angle for the other stage or stages are given from launch to burnout (approximately 900 sec). The two trajectories (figs. 1(b) and 1(c)), with negative flight-path angles are so-called dip trajectories. In order to obtain these dip trajectories the Saturn boosters were allowed to make gravity turns soon after ground launch.

Assumptions

The limit of the effective atmosphere is assumed to be 60 miles above the earth. Beyond this limit the abort trajectories of the vehicle are established by using elementary orbital mechanics (Keplerian equations, cf. ref. 1). Implicit in these equations are assumptions that (1) the vehicle experiences no drag and (2) the effects of the moon and other perturbations are neglected.

It is further assumed that at the time of abort the vehicle is separated from the booster, reoriented, and the abort velocity applied without a time delay. The launch trajectory conditions at abort, modified by any abort velocity, are then the initial conditions of the abort trajectory.

It is also assumed that the entry vehicle has a lift-to-drag ratio of 0.5. This assumption dictates the required entry conditions shown in figure 2. Such figures may be constructed from the equations of reference 2.

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Procedure

Aborts attempted at various velocities along the launch trajectories were investigated from near orbital velocity (25,000 ft/sec) to escape or burnout velocity (approximately 36,000 ft/sec) in steps of 1,000 ft/sec. In order to simplify the study, abort velocities of 1,000, 2,000, and 3,000 ft/sec were assumed. With a specific impulse I_{sp} of 260 seconds these abort velocities would represent fuel-to-vehicle mass ratios of 0.113, 0.213, and 0.301, respectively, whereas a fuel with an I_{sp} of 400 seconds would allow mass ratios of 0.075, 0.144, and 0.209, respectively. This is shown in figure 3 where the relationship between ΔV_o and $\Delta W/W$ is given by

$$\frac{\Delta W}{W} = 1 - e^{\frac{-\Delta V_o}{gI_{sp}}}$$

For a 16,000-pound vehicle, a mass ratio of 0.301 would correspondingly represent a fuel weight of 4,816 pounds.

Four methods of applying the abort velocities (illustrated in fig. 4) were investigated. In three of the methods the entire thrust was applied at the time of abort either to reduce the flight-path angle without changing the velocity as in method 1, or to reduce the velocity of the vehicle without changing the flight-path angle as in method 2, or to achieve a maximum reduction in the tangential (local horizontal) component of velocity as in method 3. In the fourth method the vehicle, after separation from the booster, is allowed to coast to apogee, and then a small velocity is applied tangent to the flight path just sufficient to insure an acceptable atmospheric entry.

The abort trajectories were specified in terms of the apogee, perigee, and (if entry occurred) the entry conditions of V_e and γ_e as defined at an altitude of 60 miles. The flight time, from abort to entry, was also determined in some cases. In the case of the trajectory of figure 1(c) aborts between orbital and escape velocities may occur within the atmosphere ($h_o < 60$ miles). In these instances, the trajectory conditions at abort (modified by any abort velocity) are the entry conditions unless the vehicle skips out of the atmosphere, in which case the reentry conditions following the first orbital period are given.

In cases in which a velocity correction was applied at apogee of the abort trajectory, the required velocity change was established by specifying the orbital perigee at 40 statute miles above the earth L
(211,200 feet). The reason for this was simplicity of calculations. 1
In so doing the entry conditions at 60 miles fell within the boundaries 4
of figure 2. 9
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In addition to the aborts initiated at superorbital velocities, calculations were made for a number of aborts initiated during the suborbital portion of the launch trajectories for the most critical cases which involve atmospheric entry (thus excluding aborts at high dynamic pressure and aborts off-the-pad). When aborts were initiated at suborbital velocities, the trajectories were computed from abort to entry with drag ($C_D = 1.4$) taken into account. At entry a lift-to-drag ratio of 0.5 was assumed for the vehicle and the trajectory was computed (eqs. of ref. 3) until the rate of descent or the altitude reached zero, whichever occurred first. For these calculations an effective drag area of 75 square feet and a vehicle weight of 7,000 pounds were used.

RESULTS AND DISCUSSION

Aborts Initiated at Superorbital Velocities

General.- Detailed results are presented only for the Saturn 3-stage dip trajectory denoted on figure 1 as trajectory 3. Limited results are presented for the Saturn 4-stage trajectories denoted as trajectories 1 and 2 on figure 1. In the detailed analysis on trajectory 3, the consequences of applying 1,000, 2,000, and 3,000 ft/sec of abort velocity when the vehicle is at a velocity of 25,000 ft/sec and at each interval of 1,000 ft/sec velocity thereafter up to 36,000 ft/sec were investigated and the results tabulated in table I.

In table I(a), the ΔV_o was applied by method 1. (See fig. 4.) At each flight condition the direction in which the abort velocity was applied α, the flight-path angle after application γ_1, the apogee h_a and perigee h_p altitudes with reference to the surface of the earth, and the atmospheric entry conditions at approximately 60 miles altitude are listed. Also given is the elapsed time Δt from abort to entry.

In table I(b) are listed the results when the ΔV_o was applied by method 2. Shown in the table are the perigee and apogee altitudes referenced to the surface of the earth, the entry conditions at 60 miles altitude, and the time of flight from abort to entry.

Similar analyses were also made on trajectories 1 and 2, but the detailed results are not tabulated herein since, for the most part, they are redundant. However, some of the more significant results are shown plotted in the figures.

<u>Aborts initiated within the recovery boundary</u>.- If abort occurs between orbital and escape velocities, the most desirable procedure may be an immediate return to the earth's atmosphere. However, for any given launch trajectory and a specified available abort velocity, immediate entry with acceptable entry conditions can be obtained only up to a certain position on the launch trajectory. This is shown in figure 5(a) by plotting the entry conditions of table I(a) on the boundary plot of figure 2. With 1,000, 2,000, and 3,000 ft/sec abort velocity available, the last position at which the vehicle may make an immediate return to earth is indicated in figure 5(a) by the point at which the skip boundary is crossed. These points for all three trajectories are marked at the bottom of each time history in figure 1. It is noted from the spacing of the recovery boundaries that for each second of delay in abort time an additional abort velocity of about 30 ft/sec would be required to effect an immediate entry. The procedure used for applying the ΔV_o is essentially that of changing the flight-path angle to a more negative value and is referred to as method 1. With the dip trajectories 2 and 3 of figure 1, an immediate entry may also be obtained without applying any abort velocity; the last position at which such a recovery can occur is also indicated on these two trajectories by a zero boundary. These positions on the launch trajectories represent the recovery boundaries for an immediate entry with the use of the specified abort velocity. For aborts initiated beyond these positions either more abort velocity is required or the abort velocity must be applied by some other method to assure eventual return to earth.

The ability to return immediately to earth by use of method 2 was also studied and the results, obtained from table I(b), are shown in figure 5(b). It may be seen that method 1 provided immediate return up to higher values of V_o and, thus, appears to be more desirable than method 2 in this respect.

<u>Aborts initiated beyond the recovery boundary</u>.- It may not be possible or desirable to enter the atmosphere immediately following abort

by applying the abort velocity by method 1. Hence, the techniques of methods 2 and 3 were investigated: the application of the entire ΔV_O along the trajectory in a retrograde direction (method 2) or tangent to the local horizon (method 3). In these cases, the vehicle will coast to an apogee and, hopefully, achieve entry near perigee.

The entry conditions actually achieved by method 2 are tabulated in table I(b) and are shown in figure 5(b) for trajectory 3. Results of method 2 similar to those shown in figure 5(b) were also obtained for the other two launch trajectories. In these cases, however, the entry conditions were acceptable only at near satellite velocities. When aborts occurred near escape velocity, the perigee of the abort trajectory was usually outside of the atmosphere or if entry did occur the entry conditions were usually unacceptable. (See fig. 2.)

One of the effects of applying the available abort velocity in a retrograde direction (method 2) is shown in figure 6. For each of the three trajectories, the apogee altitudes achieved for given values of V_O are plotted as a function of the abort velocity applied. It may be seen that the apogee altitude is not too sensitive to the amount of abort velocity until the launch velocity approaches that required for escape.

The significance of applying the abort velocity by methods 1 and 2 in terms of flight time following abort is shown in figure 7. In figures 7(a) and 7(b) if no retrovelocity is applied at the time of abort, the time from abort to entry is indicated by the solid lines. If, however, a value of ΔV_O of 1,000, 2,000, or 3,000 ft/sec is applied by method 1, the vertical lines in figure 7(a) indicate minimum abort velocity requirements for immediate entry. For example, up to a launch velocity of 28,050 ft/sec on trajectory 3, entry is achieved within seconds after abort without any abort velocity. Above this launch velocity of 28,050 ft/sec, immediate capture will not be achieved and an orbital period of at least 2 hours will occur before reentry and possible capture on the return pass. If a value of 1,000 ft/sec of ΔV_O can be applied by method 1, then entry and capture within seconds after abort may be accomplished up to a launch velocity of 30,500 ft/sec. At higher launch velocities, the vehicle will not be captured during its initial pass through the atmosphere and will have an orbital period of 3 hours or more before its next pass and possible capture within the atmosphere. When aborts occur at velocities greater than 34,050 ft/sec on trajectory 3, orbital periods will be at least 11 hours unless abort velocities greater than 3,000 ft/sec are applied by method 1 to achieve immediate capture or unless the vehicle velocity is reduced by some method at the time of abort.

The effect on the flight time of using the available ΔV_O to achieve a maximum reduction in the vehicle velocity (method 2) is shown in figure 7(b). With a value of ΔV_o of 1,000 ft/sec, entry and capture within seconds after abort are possible up to V_o = 29,050 ft/sec. For aborts initiated beyond V_o = 29,050 ft/sec, skip-out will occur with orbital periods of at least 2 hours as indicated by the dashed line marked ΔV_o = 1,000 ft/sec. These results (for launch trajectory 3) indicate that, at the time of injection ($V_o \rightarrow$ 36,000 ft/sec), abort with 3,000 ft/sec applied along the flight path (method 2) will place the abort vehicle into an orbit of 6.7 hours duration with an apogee of 14,500 miles (fig. 6) from the earth. It may also be seen that, beyond the recovery boundary, method 2 provides an earlier return than does method 1.

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It has been suggested that if the available ΔV_o is applied tangent to the local horizon (method 3), the time until entry will be minimized. Just how much time is saved was investigated by comparing the times from abort to entry when the ΔV_o was applied by methods 2 and 3. Trajectory 1 was used for this comparison since it afforded the largest injection flight-path angle of the three trajectories studied and hence the largest difference in the direction of application of ΔV_o by the two methods. The results, for aborts at a V_o = 35,000 ft/sec, are shown in figure 8. The results show that the apogee altitude and the elapsed time from abort to entry are virtually unchanged. In fact, applying the thrust by method 3 gave slightly longer return times. The time to return was found to be extremely sensitive to the reduction in flight velocity ($V_o - \Delta V_o$ for method 2 and $V_o - \Delta V_o \cos \gamma_o$ for method 3), and this difference in velocity between the two methods more than compensated for the changes in return time due to changing the flight-path angle. A more significant observation is that the time from abort to entry is very insensitive to small errors in the direction in which the retrograde abort velocity is applied. Thus, the accuracy requirements of the angles of applying the abort velocity for aborts beyond the recovery boundary may not be too stringent from time-to-return considerations.

Velocity corrections at apogee.- It was pointed out previously that in many cases in which aborts were initiated at superorbital velocities the perigee of the abort trajectory either did not lie within the earth's atmosphere or if it did, the entry angle was too low for capture. In no case examined, where abort was initiated at superorbital velocities, was the entry angle larger than that specified by the heat boundary of figure 2. However, if abort is initiated due to a significant deviation of the booster from its nominal trajectory, an excessively large entry angle may also be incurred.

It appears, therefore, that regardless of what action is taken at the time of abort, some provision for making further velocity corrections near apogee will be required to insure that the entry conditions are within the specified limits of the vehicle. If the apogee distance is so large that the corrections made at apogee are not sufficiently accurate to insure the proper entry conditions, additional fuel may also be required for midcourse corrections during the return phase of the trajectory.

In order to study the magnitude of velocity corrections at apogee to insure entry, the vehicle was simply allowed to coast to apogee without any abort velocity being applied at initiation of abort. The perigee altitude of this orbit was then calculated. Of the three launch trajectories shown in figure 1, aborts at superorbital velocities from trajectory 1 led to the highest perigee altitudes (earth miss distances). The velocity changes at apogee ΔV_a required to lower the perigee to the value necessary for capture are shown in figure 9. Also plotted as a function of the vehicle velocity at the time of abort V_o are variations of apogee altitude and time of flight from abort to entry. It may be seen in figure 9 that the largest velocity correction of 210 ft/sec was required for aborts at near-orbital velocities. As V_o increased, the apogee altitude and flight time increased, while the velocity correction at apogee decreased.

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In addition to these corrections at apogee, some subsequent midcourse velocity corrections will also be required. Because of errors inherent to the navigational equipment, the accuracy of predicting the perigee altitude and hence computing exactly what velocity corrections to make at apogee will decrease with apogee distance. For instance, at the greatest apogee distance shown on figure 9, 60,000 miles, perigee distance can be ascertained only to within a corridor of width somewhere between 80 and 360 miles (depending upon the source of the navigational data) if no other corrections were made. Reference 4 indicates that between 100 and 200 ft/sec of corrective velocity will be required during the approach to earth to insure the desired entry conditions.

Aborts Initiated at Suborbital Velocities

In addition to the foregoing analysis on aborts initiated at superorbital velocities, a limited analysis was made of aborts initiated at suborbital velocities. The study was limited to abort trajectories which extended beyond the 60-mile altitude prior to an atmospheric entry. Such abort trajectories may first be incurred when the booster and payload are still in the atmosphere and the dynamic pressure has decreased from a maximum to less than about 10 lb/sq ft. Aborts at these conditions are usually characterized by moderately high (10° to 30°) flight-path angles, and low (5,000 to 6,000 ft/sec) velocities both at the time of abort and at entry. A typical abort trajectory at this condition is shown in figure 10(a). The abort was initiated at $t = 173$ seconds on launch trajectory 3 of figure 1. Since the abort occurred at an altitude of 47 miles, V and $|\gamma|$ were about the same at abort and where the deceleration started to build up on entry. With an L/D of 0.5, the vehicle experienced a maximum deceleration of 6g without abort thrust except that required for the initial separation from the booster.

For aborts initiated after the booster and vehicle leave the atmosphere, the entry conditions become more critical than those illustrated by figure 10(a). Although the flight-path angle is decreasing with time, the corresponding increases in velocity and altitude seriously affect the entry conditions. Of the launch trajectories investigated, the worst (suborbital) entry conditions were obtained when the aborts were initiated at launch velocities near 16,000 ft/sec. A typical result is shown in figure 10(b). This trajectory was obtained when an abort was initiated at $t = 305$ seconds on launch trajectory 1 of figure 1. Only the entry phase of the abort trajectory is shown although the time required from abort to apogee and back to the original abort altitude was about 200 seconds and the apogee altitude was about 150 statute miles.

The trajectory of figure 10(b) shows the importance of abort altitude and launch velocity on the entry conditions for aborts initiated at suborbital velocities. Although the value of the flight-path angle at abort was only 10°, the entry flight-path angle was -20°. Even with an L/D of 0.5, this combination of velocity and flight-path angle led to a maximum deceleration of 18g.

The reason for the large absolute change in flight-path angle is directly due to the 100 miles difference in altitude (between abort and entry) and the large difference (0.7g) between the gravitational and the centrifugal accelerations. In order to further correlate the conditions at the abort altitude with the conditions at the entry altitude, a variation of γ with V at these two conditions is shown in figure 11 for trajectories 1 and 3. In the figure the solid lines show the variations of the flight-path angle with velocity along the two launch trajectories. On these solid lines altitude is variable. However, for each position on the launch trajectory there is an associated entry velocity and flight-path angle which is referenced to one altitude. The dashed line gives these conditions for an altitude of 50 miles above the earth. Several corresponding points which relate the conditions at abort and entry are joined by arrows and the maximum accelerations experienced by a vehicle ($L/D = 0.5$) during the entry is indicated at the arrowhead. The figure shows that the worst entry conditions were obtained near a $V_o = 16,000$ ft/sec and that at this condition the γ_o for trajectory 1 was lower than γ_o for trajectory 3. However, due to the difference in altitude (see fig. 1) between the two trajectories at the time of abort, the change in $|\gamma|$ for trajectory 1 was 12° while the change in $|\gamma|$ for trajectory 3 was only 4.5°.

Thus, it is not always obvious what the most critical conditions for suborbital abort may be unless altitude, velocity, and flight-path angle are all taken into account or unless all velocities and flight-path angles are corrected to the appropriate entry altitude. For this latter method, the closed-form solutions to Kepler's equations (cf. ref. 1) are sufficient.

Beyond the critical condition illustrated in figures 10(b) and 11, the combination of even lower abort flight-path angles and higher centrifugal accelerations (due to higher velocities) leads to less severe conditions at entry.

The large maximum deceleration at entry illustrated in figure 10(b) may be largely alleviated by reducing the entry angle. Velocity impulses of 1,000, 2,000, and 3,000 ft/sec applied normal to the flight path just prior to entry (V_e = 15,000 ft/sec) on the trajectory of figure 10(b) produced the results shown in the following table:

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ΔV, ft/sec	$\Delta\gamma$, deg	$(\Delta a_n)_{max}$, g units	$(a_n)_{max}$, g units
0	0	0	17.6
1,000	3.8	3.1	14.5
2,000	7.6	6.1	11.5
3,000	11.3	8.2	9.4

The heating aspects of such entries have not been quantitatively investigated. However, it has been established (cf. refs. 3 and 5) that the maximum heating rate will be reduced and the total heat absorbed will be increased by reducing the entry angle just prior to entry. Since the total heat transfer is strongly dependent on the energy at entry, aborts at suborbital velocities should not be expected to exceed the hypervelocity design conditions. Therefore the trade-off of high heating rates at high accelerations for longer soak periods (higher total heat transfer) and lower accelerations should prove to be more favorable to the latter.

IMPLICATIONS OF RESULTS

The present investigation was concerned with possible methods of utilizing available fuel in the case of aborts during launch on a lunar mission. The results indicate that flight conditions during the launch change very rapidly and that, depending upon the amount of available fuel and the instantaneous flight conditions at the time of abort, the proper procedure may also change quite rapidly. Up to a certain point on the launch trajectory, somewhere between orbital and escape velocities, the available abort fuel may be used to reduce the flight-path angle and effect an immediate return to earth. Beyond that point the velocity should be applied essentially along the flight path to reduce the velocity. From the standpoint of time from abort to entry, the exact angle at which this retrograde velocity is applied is not too critical. However, the entry angle will be directly dependent upon the direction of application at the time of abort. For this reason, some abort fuel should be held in reserve for further corrections at apogee and during the return to earth to insure that proper entry conditions are attained.

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Aborts initiated during the initial exit of the vehicle from the atmosphere (suborbital velocities) may be quite critical from deceleration considerations on entry. However, large changes in flight-path angle may be achieved just prior to entry since the entry velocity for the most critical cases was only about 15,000 ft/sec.

In this analysis impulsive thrust was assumed and the time delay in initiating the thrust neglected. The analysis shows that beyond the recovery boundary the time allowed to modify the velocity and hence the abort trajectory is not critical. However, the capacity to return immediately to earth, and hence the very definition of the recovery boundary is strongly influenced by the time required to initiate and apply the abort velocity. At near-escape velocities the impulsive velocity requirement for an immediate return to earth increases at a rate of about 30 ft/sec every second regardless of whether the vehicle is being boosted $\left(\text{at about } 1\frac{1}{2} \text{ to } 2g\right)$ or is in free flight after separation from the booster. The variables most important to the abort requirements (γ, $\dot{\gamma}$, and h) are virtually unaffected by the relatively small changes in velocity from accelerated or unaccelerated flight. Thus, the time delay in applying the abort velocity will be very critical in this region.

The actual time required to apply the thrust (and hence the choice of the total impulse and thrust level of the rocket engine) will strongly affect the position of the recovery boundary. As the vehicle approaches escape velocity, the centrifugal acceleration exceeds the gravitational accelerations by as much as 1.0g and this difference in acceleration produces a positive time rate of change of the flight-path angle. In order to return to the earth immediately, this acceleration difference must be temporarily overcome and the flight-path angle reduced to a sufficiently large negative value to bring the vehicle back into the atmosphere within the boundaries defined in figure 2. The significance of finite burning time can be illustrated with the following numerical example.

A vehicle having an (initial) 1g abort rocket engine with a specific impulse of 400 seconds would require 82 seconds to apply a 3,000 ft/sec velocity change and this velocity change would consume 20.5 percent of the vehicle weight. Thus, if this abort velocity were applied essentially perpendicular to the flight path when the vehicle is at near-escape velocities, then for 82 seconds, the vehicle acceleration due to the abort rocket thrust would serve only to counteract the centrifugal acceleration and would not significantly reduce the altitude.

At suborbital velocities it was pointed out that abort velocities applied impulsively just prior to entry could significantly change the flight-path angle. However, the magnitude of this change may be reduced by finite firing times and this loss of effectiveness may lead to less favorable deceleration conditions.

In any complete analysis, it is concluded that the time required to separate the vehicle from the booster, the time required to orient the vehicle in the correct direction to apply the abort velocity, and the time required to effect a velocity change will strongly influence the results of this preliminary analysis based on impulsive thrust. It is also concluded that the changes in the results due to these factors will generally be unfavorable from considerations such as the total velocity requirements, the entry conditions, and the size of rocket engine used for the abort.

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Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., February 21, 1961.

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TABLE I.- ABORT TRAJECTORIES AND ENTRY CONDITIONS

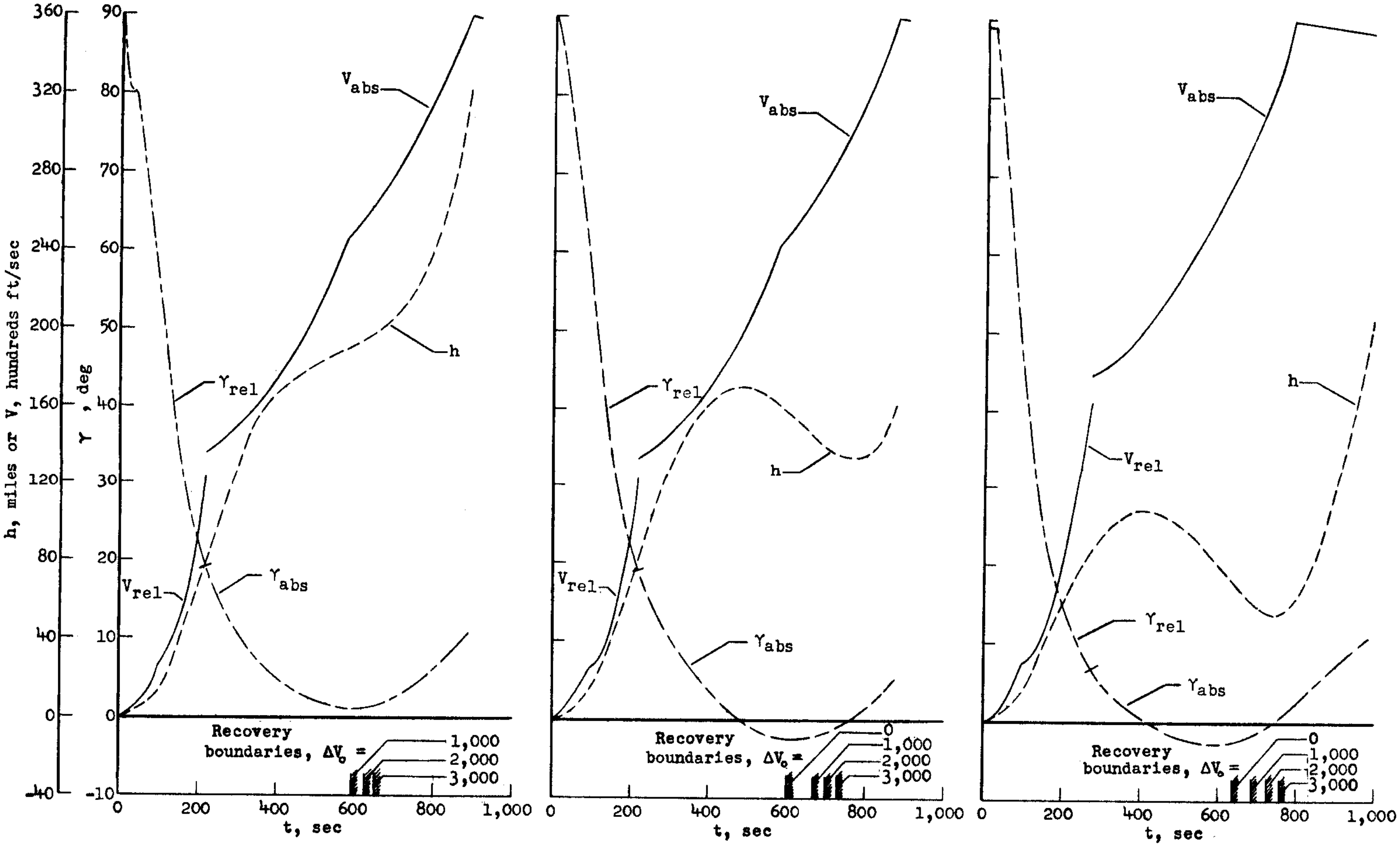
[Launch trajectory 3]

(a) ΔV_o applied by method 1

Conditions at time of abort			ΔV_o = 1,000 ft/sec								ΔV_o = 2,000 ft/sec								ΔV_o = 3,000 ft/sec							
							Entry conditions								Entry conditions								Entry conditions			
V_o, ft/sec	h_o, miles	γ_o, deg	α, deg	γ_1, deg	h_a, miles	h_p, miles	h_e, miles	V_e, ft/sec	γ_e, deg	Δt, hr	α, deg	γ_1, deg	h_a, miles	h_p, miles	h_e, miles	V_e, ft/sec	γ_e, deg	Δt, hr	α, deg	γ_1, deg	h_a, miles	h_p, miles	h_e, miles	V_e, ft/sec	γ_e, deg	Δt, hr
25,000	88.5	-2.5	88.9	-4.79	269	-484	60	25,186	-5.03	0.019	87.7	-7.09	406	-621	60	25,186	-7.24	0.013	86.6	-9.38	549	-765	60	25,186	-9.50	0.010
26,000	82.0	-2.8	88.9	-5.00	561	-191	↓	26,139	-4.91	.014	87.8	-7.21	715	-345	↓	26,139	-7.14	.010	86.7	-9.41	873	-502	↓	26,140	-9.38	.008
27,000	75.5	-2.7	88.9	-4.82	1,140	-58	↓	27,094	-4.57	.010	87.9	-6.95	1,255	-153	↓	27,094	-6.77	.007	86.8	-9.07	1,386	-304	↓	27,094	-8.94	.005
28,000	70.0	-2.4	89.0	-4.45	1,975	-5	↓	28,059	-4.15	.007	88.0	-6.49	2,053	-82	↓	28,059	-6.30	.004	86.9	-8.54	2,151	-181	↓	28,059	-8.39	.004
29,000	65.0	-2.1	89.0	-4.08	3,084	18	↓	29,028	-3.86	.004	88.0	-6.05	3,139	-37	↓	29,028	-5.91	.002	87.0	-8.03	3,212	-110	↓	29,028	-7.92	.002
30,000	61.5	-1.8	89.0	-3.71	4,569	30	↓	30,008	-3.62	.001	88.1	-5.62	4,609	-11	↓	30,008	-5.56	.001	87.1	-7.53	4,665	-67	↓	30,008	-7.49	0
31,000	58.0	-1.2	89.1	-3.05	6,606	40	↓	30,989	-3.20	3.508	88.2	-4.90	6,634	11	58	31,000	-4.90	0	87.2	-6.75	6,676	-30	58	31,000	-6.75	0
32,000	56.0	-.9	89.1	-2.69	9,591	43	↓	31,979	-3.09	4.641	88.2	-4.48	9,613	21	56	32,000	-4.48	0	87.3	-6.27	9,646	-12	56	32,000	-6.27	0
33,000	55.0	-.2	89.1	-1.94	14,331	49	↓	32,974	-2.62	6.645	88.3	-3.67	14,344	34	60	32,974	-4.06	6.645	87.4	-5.41	14,374	9	55	33,000	-5.41	0
34,000	55.0	.4	89.2	-1.29	23,014	53	↓	33,976	-2.27	10.884	88.3	-2.97	23,024	42	↓	33,976	-3.51	10.884	87.5	-4.66	23,043	24	55	34,000	-4.66	0
35,000	56.0	1.0	89.2	-.64	43,919	55	↓	34,981	-1.84	23.589	88.4	-2.27	43,926	49	↓	34,981	-2.51	23.589	87.5	-3.91	43,939	36	60	34,981	-4.28	23.589
36,000	58.5	1.8	89.2	.21	164,444	58	↓	35,993	-1.11	142.862	88.4	-1.38	164,448	56	↓	35,993	-1.76	142.862	87.6	-2.98	164,454	47	60	35,993	-3.17	142.862

(b) ΔV_o applied by method 2

Conditions at time of abort			ΔV_o = 0						ΔV_o = 1,000 ft/sec						ΔV_o = 2,000 ft/sec						ΔV_o = 3,000 ft/sec					
					Entry conditions						Entry conditions						Entry conditions						Entry conditions			
V_o, ft/sec	h_o, miles	γ_o, deg	h_a, miles	h_p, miles	h_e, miles	V_e, ft/sec	γ_e, deg	Δt, hr	h_a, miles	h_p, miles	h_e, miles	V_e, ft/sec	γ_e, deg	Δt, hr	h_a, miles	h_p, miles	h_e, miles	V_e, ft/sec	γ_e, deg	Δt, hr	h_a, miles	h_p, miles	h_e, miles	V_e, ft/sec	γ_e, deg	Δt, hr
25,000	88.5	-2.5	145	-369	60	25,188	-3.00	0.039	115	-839	60	24,194	-3.56	0.036	104	-1,258	60	23,202	-4.16	0.031	99	-1,619	60	22,211	-4.75	0.030
26,000	82.0	-2.8	412	-42	↓	26,139	-2.63	.026	155	-395	↓	25,144	-3.12	.025	114	-860	↓	24,150	-3.59	.024	101	-1,273	↓	23,156	-4.05	.023
27,000	75.5	-2.7	1,052	30	↓	27,094	-2.21	.020	388	-46	↓	26,098	-2.58	.019	143	-405	↓	25,102	-2.94	.018	105	-871	↓	24,106	-3.31	.018
28,000	70.0	-2.4	1,923	48	↓	28,059	-1.79	.014	1,021	33	↓	27,061	-2.06	.014	714	-35	↓	26,063	-2.32	.014	124	-396	↓	25,065	-2.59	.013
29,000	65.0	-2.1	3,050	52	↓	29,028	-1.63	2.309	1,892	48	↓	28,030	-1.77	.008	994	36	↓	27,030	-1.91	.008	322	-23	↓	26,032	-2.06	.008
30,000	61.5	-1.8	4,549	53	↓	30,010	-1.73	2.796	3,024	52	↓	29,009	-1.65	2.301	1,871	49	↓	28,009	-1.69	.003	973	40	↓	27,009	-1.73	.002
31,000	58.0	-1.2	6,590	55	↓	30,989	-1.56	3.508	4,510	55	↓	29,989	-1.51	2.784	2,996	54	↓	28,989	-1.46	2.293	1,846	52	↓	27,988	-1.39	1.941
32,000	56.0	-.9	9,580	55	↓	31,979	-1.76	4.641	6,575	54	↓	30,979	-1.67	3.498	4,493	54	↓	29,978	-1.59	2.777	2,981	54	↓	28,977	-1.48	2.288
33,000	55.0	-.2	14,328	55	↓	32,975	-1.80	6.646	9,561	55	↓	31,974	-1.68	4.633	6,541	55	↓	30,973	-1.60	3.493	4,481	55	↓	29,973	-1.54	2.775
34,000	55.0	.4	23,012	55	↓	33,976	-1.91	10.883	14,328	55	↓	32,975	-1.83	6.646	9,561	55	↓	31,974	-1.74	4.633	6,551	55	↓	30,973	-1.64	3.493
35,000	56.0	1.0	43,920	55	↓	34,981	-2.00	23.589	23,072	55	↓	33,981	-1.96	10.915	14,360	54	↓	32,978	-1.88	6.660	9,580	54	↓	31,979	-1.82	4.641
36,000	58.5	1.8	164,449	54	↓	35,993	-2.11	142.862	44,341	54	↓	34,993	-2.09	23.876	23,223	54	↓	33,993	-2.06	10.995	14,438	53	↓	32,992	-2.06	6.695



(a) Trajectory 1. (b) Trajectory 2. (c) Trajectory 3.

Figure 1.- Typical launch trajectories for circumlunar missions with Saturn boosters.

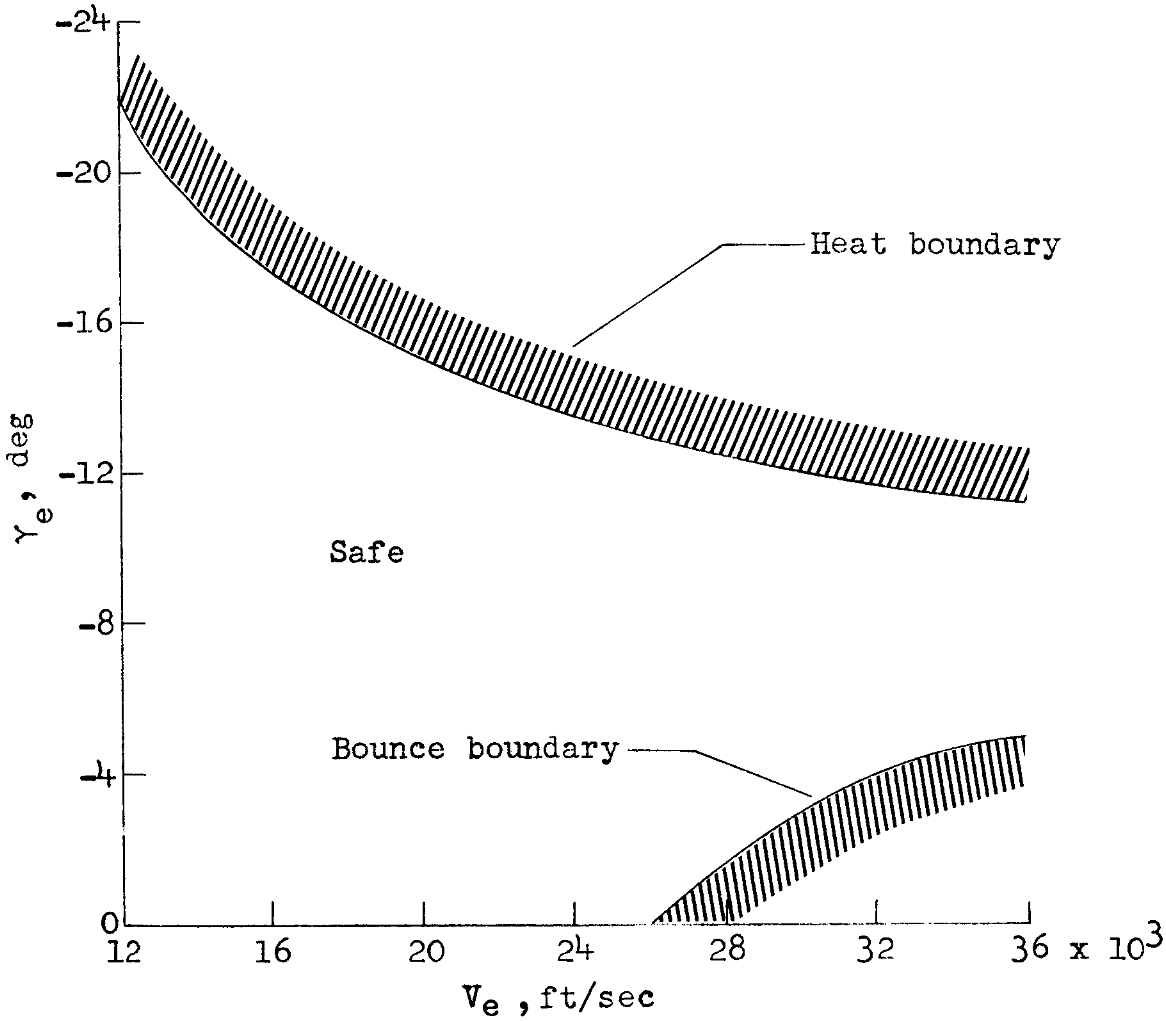


Figure 2.- Required entry conditions.

L-1493

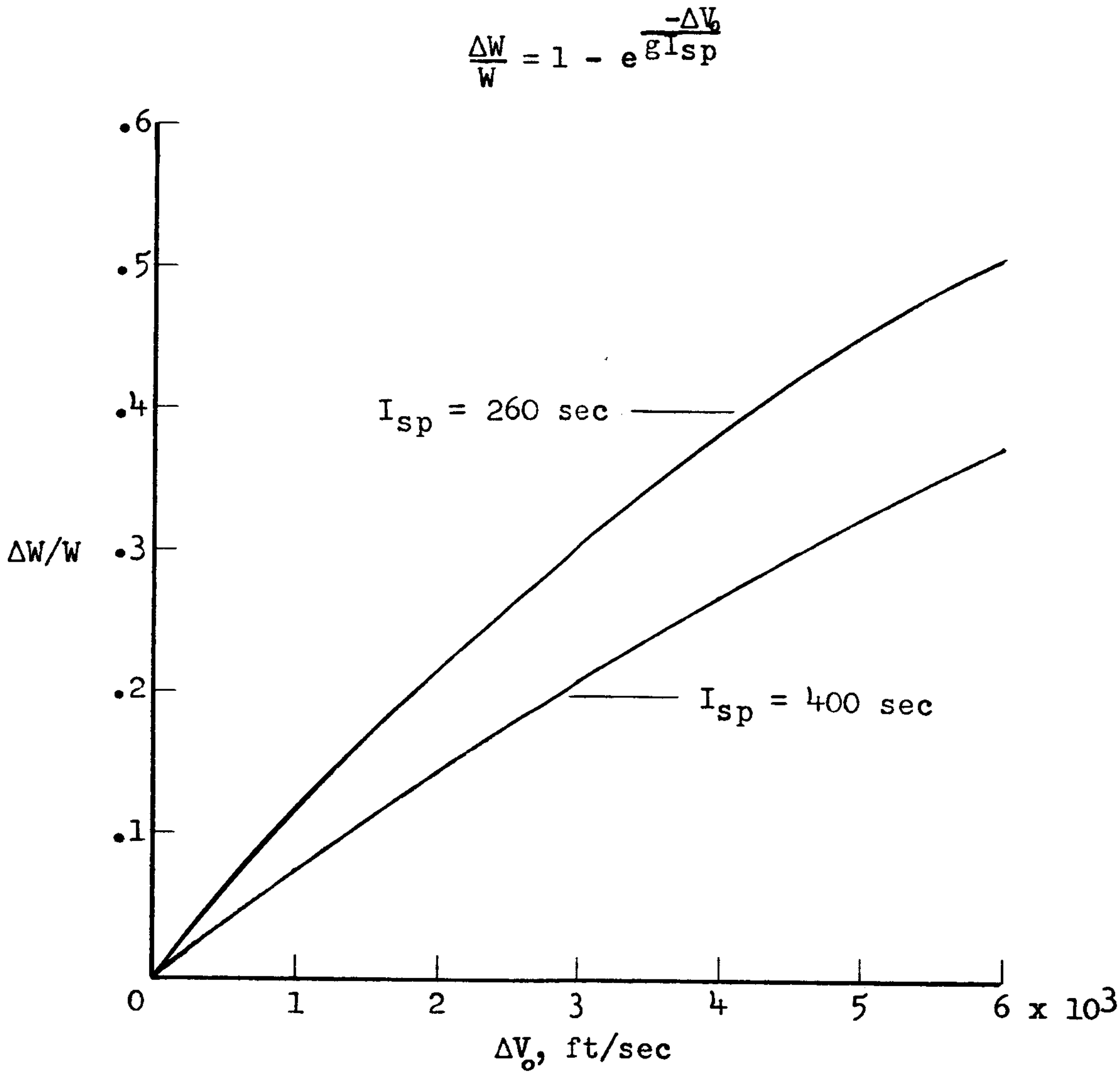
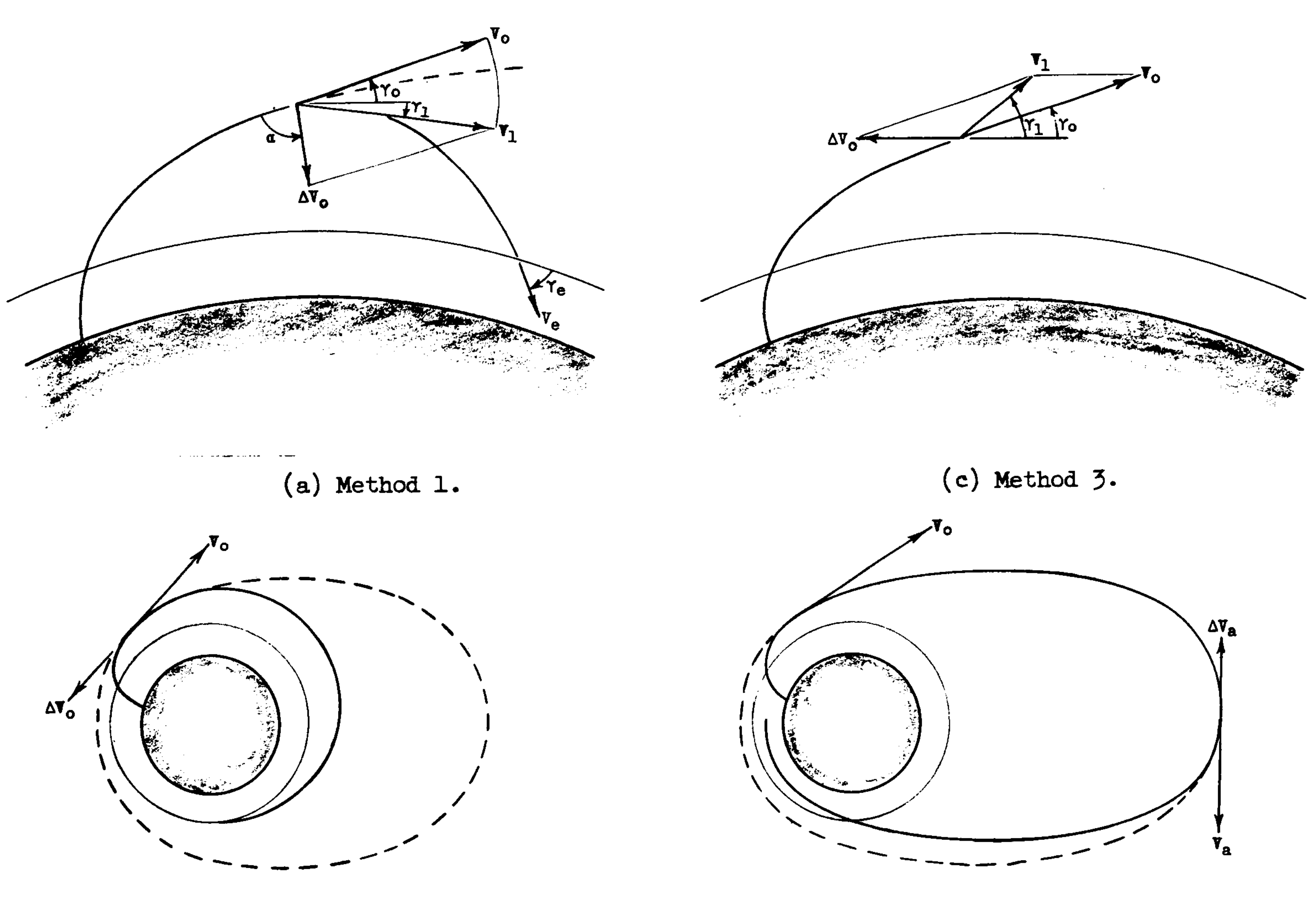


Figure 3.- Relationship between abort velocity and ratio of fuel weight to vehicle weight.



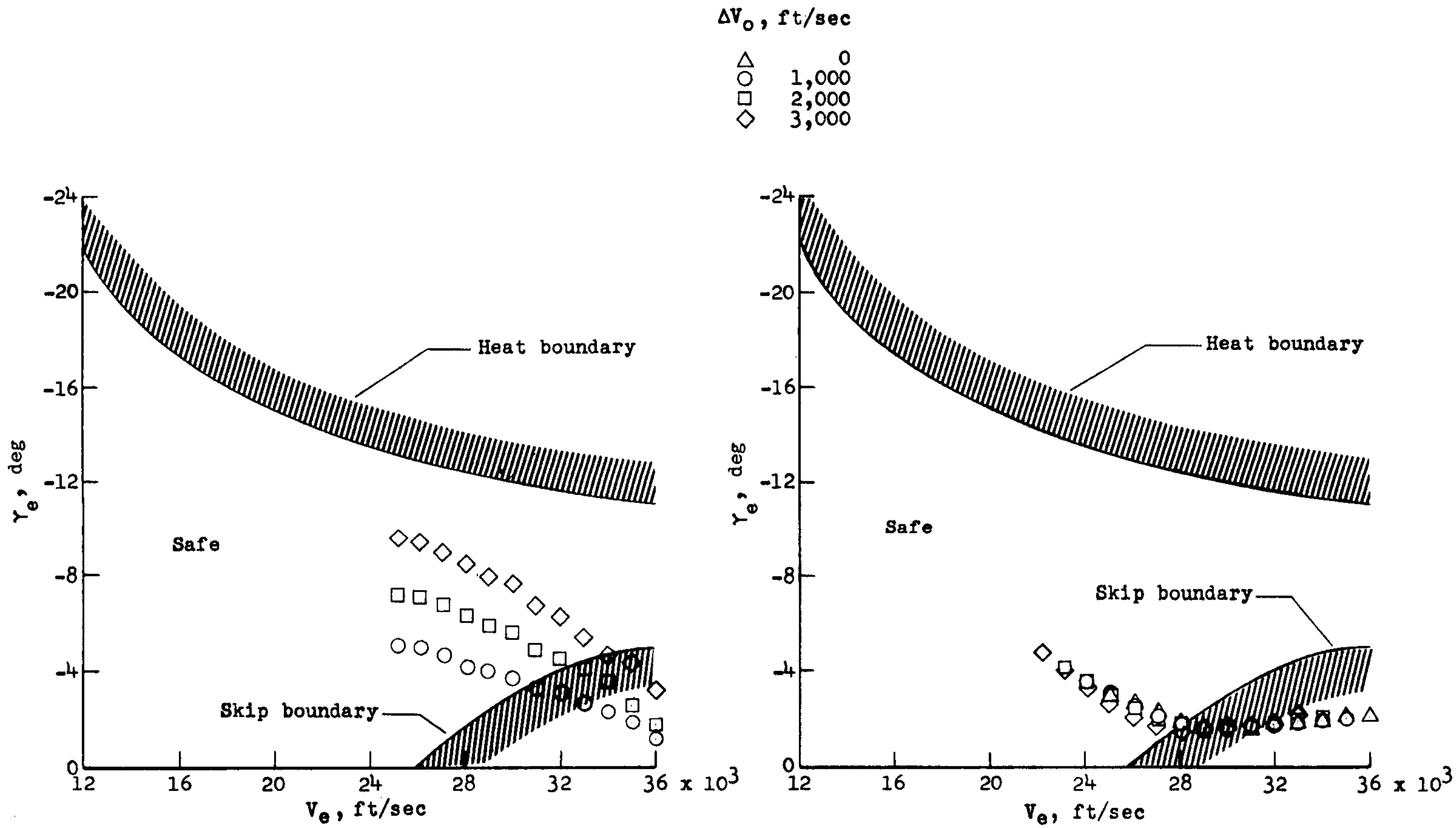
(a) Method 1.

(c) Method 3.

(b) Method 2.

(d) Method 4.

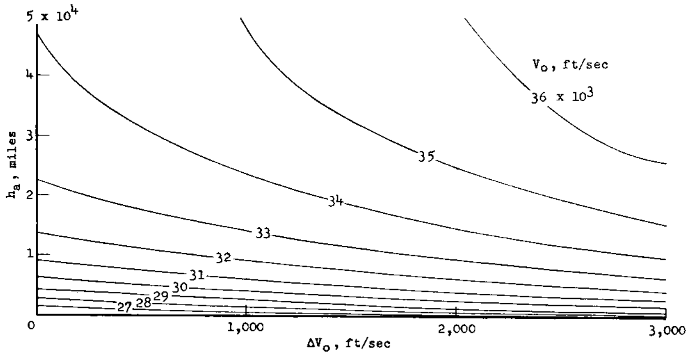
Figure 4.- Methods of applying the abort velocities.



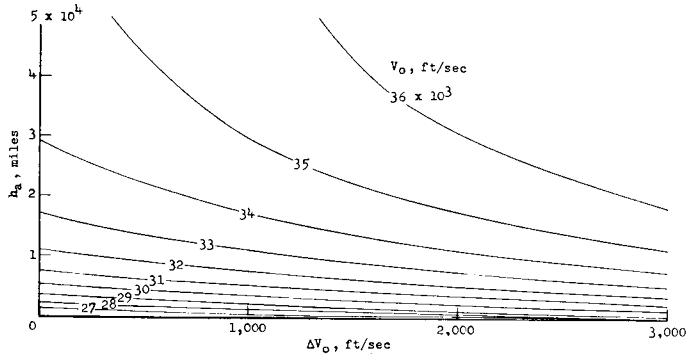
(a) ΔV_o applied by method 1.

(b) ΔV_o applied by method 2.

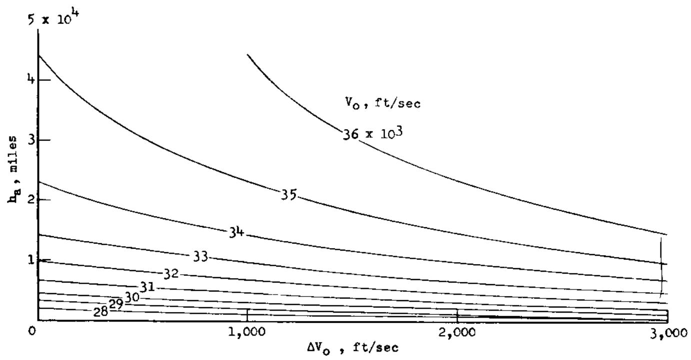
Figure 5.- Abort trajectory entry conditions. Launch trajectory 3.



(a) Trajectory 1.

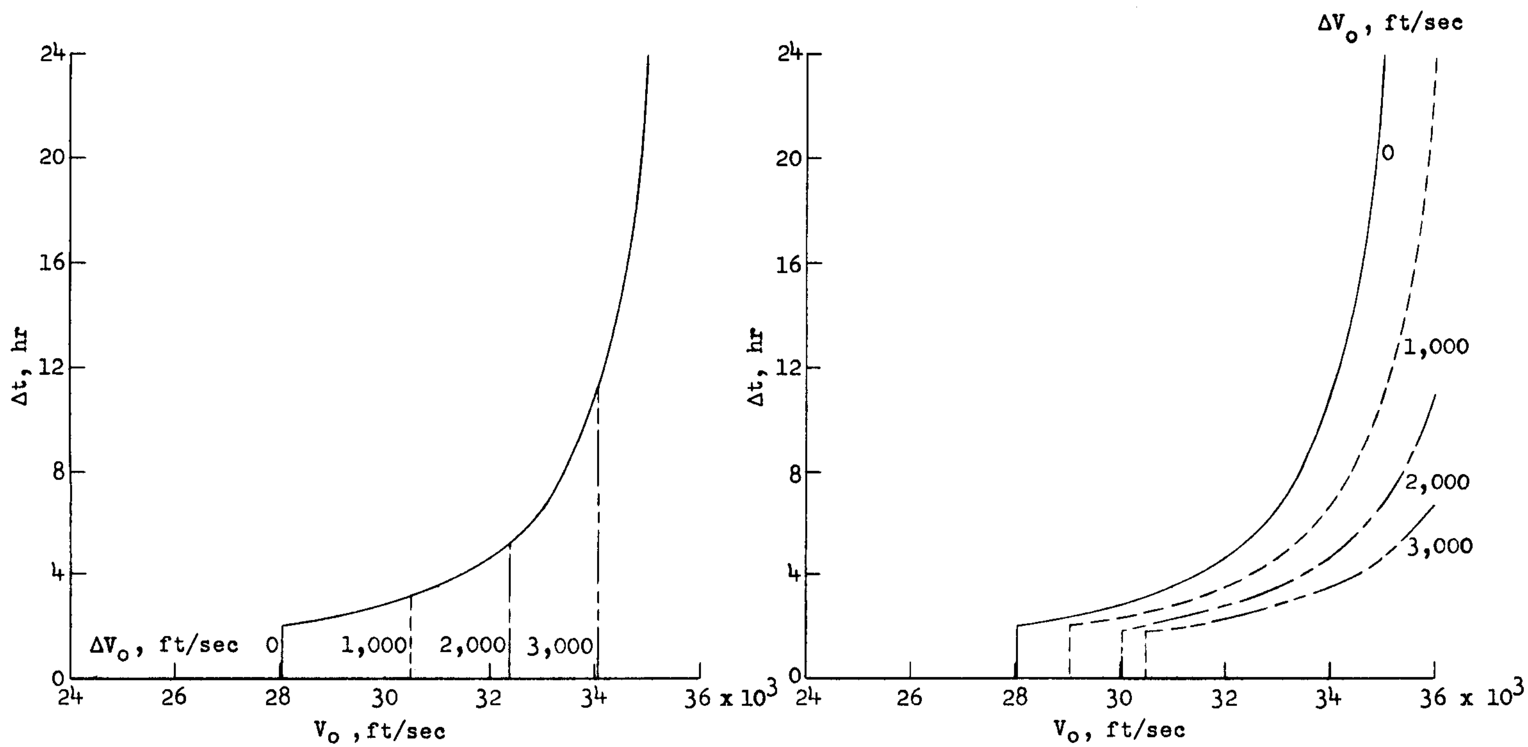


(b) Trajectory 2.



(c) Trajectory 3.

Figure 6.- Variation of apogee distance with ΔV_o, as applied by method 2.



(a) ΔV_o applied by method 1.

(b) ΔV_o applied by method 2.

Figure 7.- Flight time of vehicle from abort to entry. Trajectory 3.

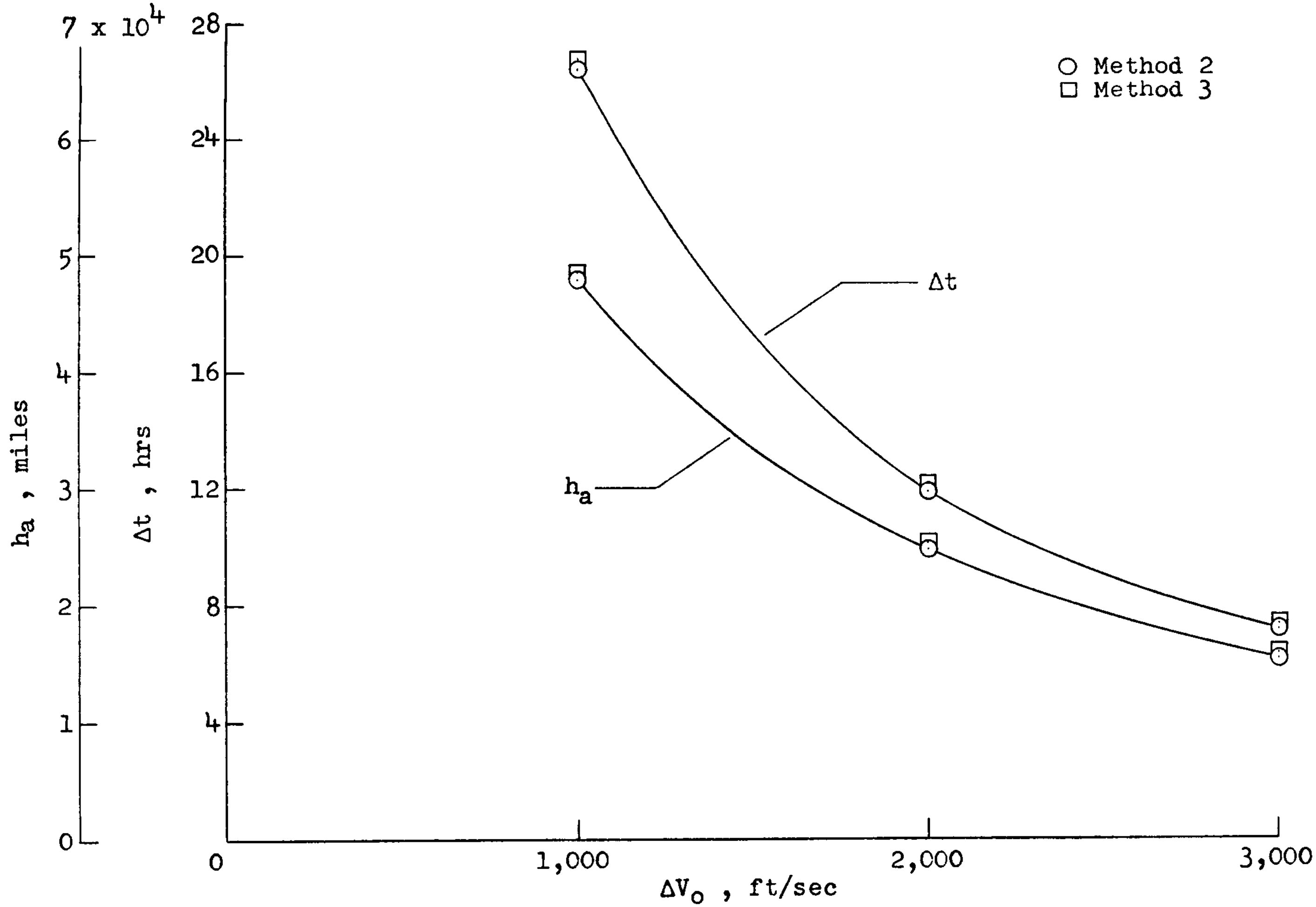


Figure 8.- Comparison of abort trajectories resulting from application of method 2 and method 3. Initial abort conditions: V_o = 35,000 ft/sec; h_o = 301 miles; $\gamma_o = 9.7^o$ of trajectory 1.

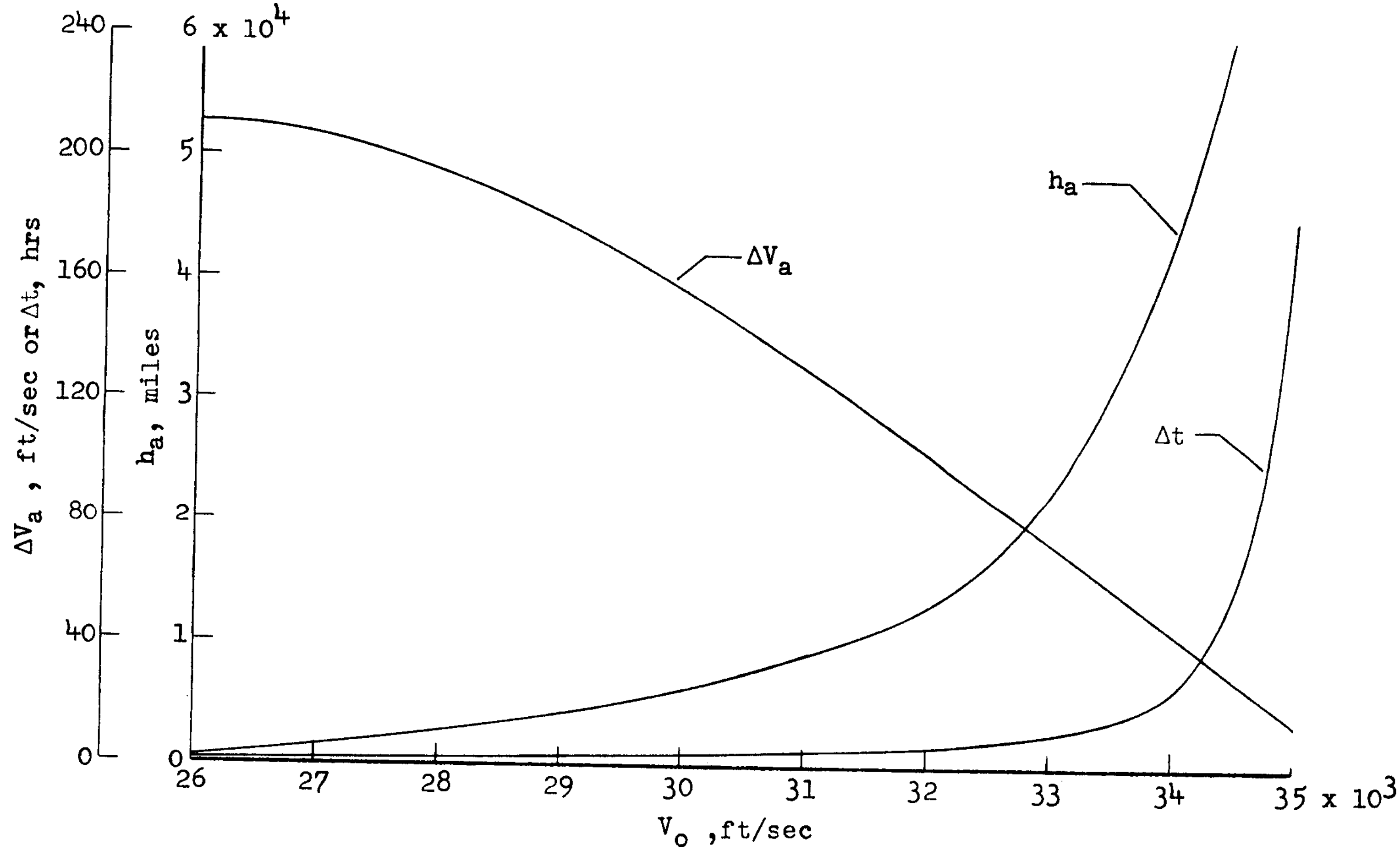
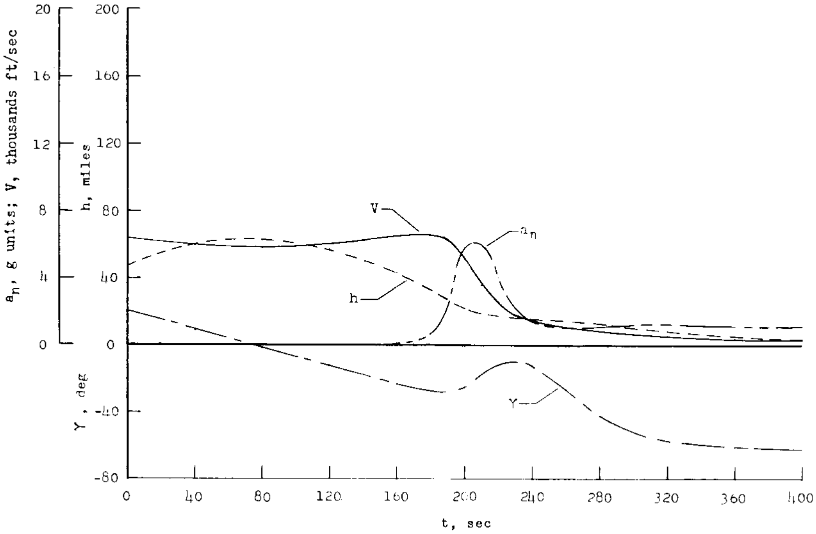
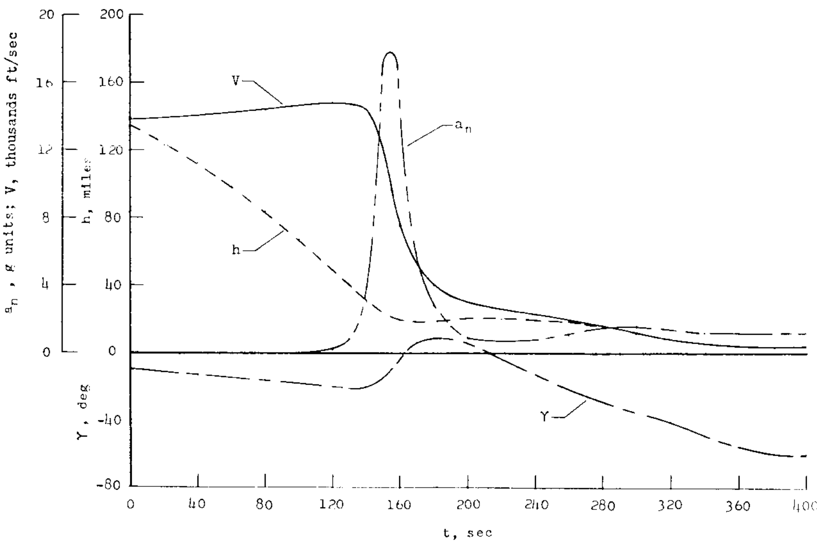


Figure 9.- Abort trajectories resulting from application of method 4. V_o of launch trajectory 1.



(a) Initial abort conditions of trajectory 3: $V_o = 6{,}400$ ft/sec; $h_o = 47$ miles; $\gamma_o = 20^o$.



(b) Entry conditions for an abort initiated during trajectory 1: $V_o = 13{,}800$ ft/sec; $h_o = 133.5$ miles; $\gamma_o = 10^o$.

Figure 10.- Abort trajectories initiated at suborbital velocities with relatively high entry angles and with $L/D = 0.5$ during entry.

L-1493

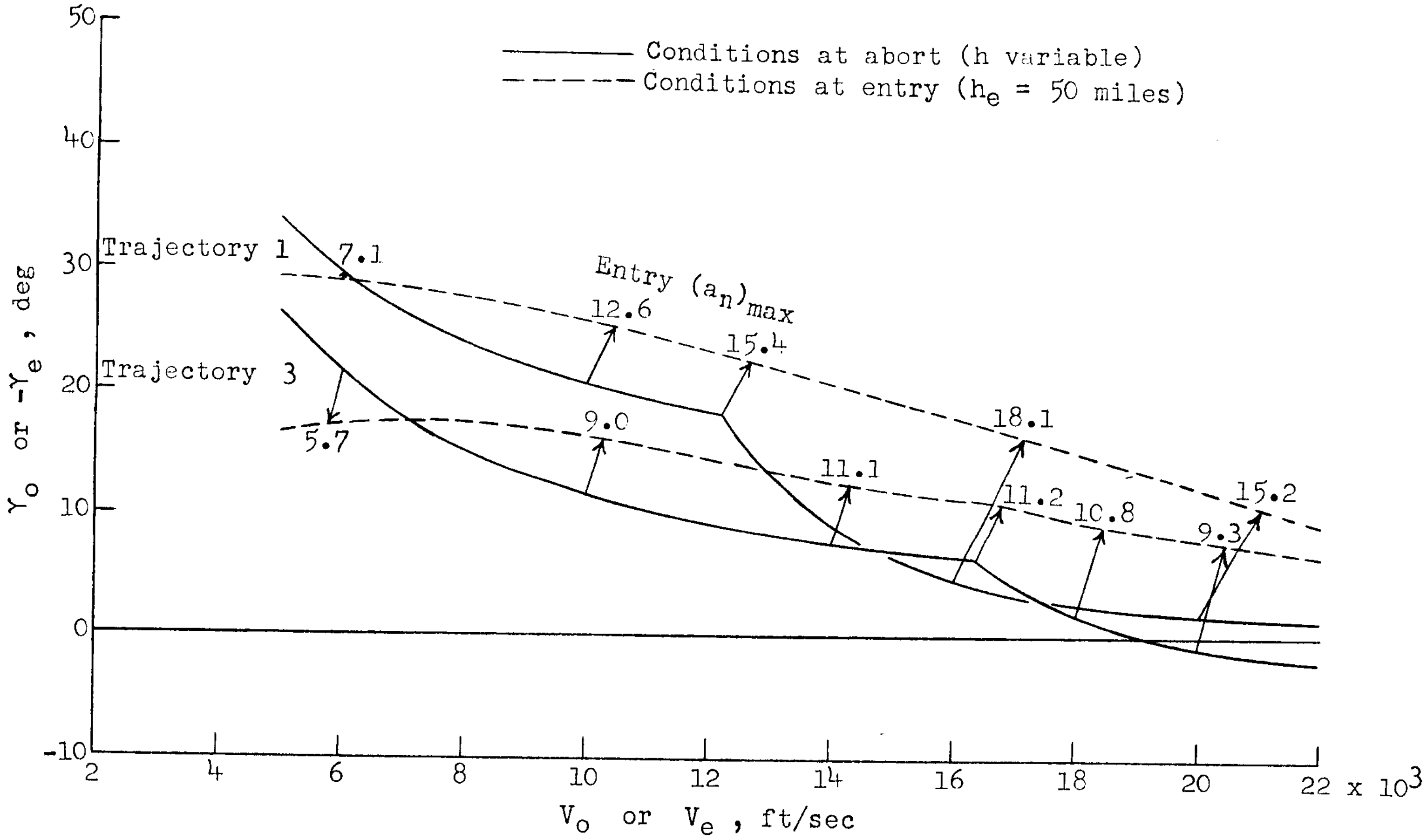


Figure 11.- A comparison of the velocities and flight-path angles between abort and entry for trajectories 1 and 3. Maximum entry decelerations at several conditions are listed.

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